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TOPEX/POSEIDON Ground Track Maintenance Experience*

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The TOPEX/POSEIDON satellite is maintained in a nearly circular, frozen orbit ($e \approx 0.000095$, ω: 90°) at an altitude of ≈1336 km, and an inclination of $i \approx 66.04°$, which provides an exact repeat ground track every 127 revolutions (≈9.9 days) and overflies two altimetry verification silts. Orbit maintenance maneuvers are required to recover orbital decay due to drag **arid** ground track drift due to luni-solar gravity. Satellite fixed forces, which may include outgasing or fuel leakage, produce a continuous thrust on the order of micro newtons, and constitute the largest uncertainty to maneuver design. Maneuver targeting strategies were redesigned in flight to incorporate the effects of this unexpected perturbation. These new targeting strategies are currently being used to design and implement ground track maintenance maneuvers. Oth er tots include drag prediction uncertainty, maneuver execution errors, and orbit determination errors. Maneuver targeting incorporates all of these errors. Since acquiring the operational orbit on Sept. 25, 1992, the ground track has been maintained with $\mu \approx 0.15\,km$ and $\sigma \approx 0.30\,km$. Over W% of the more than 1600 nodal crossings which occurred during this time have been within the ±1 km reference bandwidth, well exceeding mission requirements.*** This paper will describe our success at maintaining the operational ground track. The maintenance of the ground track al the ascending nodes as well as at the verification sites Will be discussed separately in some detail. Maneuvers which have been performed will be discussed, in terms of the targeting strategy used and the successes thereof. Successes and failures of the planned targeting strategies will be examined, and modifications to the maneuver targeting strategies which have been developed and implemented since launch will be described.

INTRODUCTION

TOPEX/POSEIDON was launched by an Ariane 42P on August 10, 1992 with injection occurring at 23:27:05 UTC, approximately 19 min, 57 sec after lift off. Its goal is to determine ocean surface height to an accuracy of 13 cm. (3σ) utilizing a combination of satellite altimetry data and precision orbit determination. The mission profile is organized into the following phases: assessment, initial verification, observational, and extended observational [Carlisle 1991]. Assessment involved an intense period Of satellite performance assessment and calibration, sensor initialization, and acquisition of the operational orbit. Six maneuvers were performed during the assessment phase to acquire the operational orbit. These maneuvers raised and circularized the orbit, removed inclination errors induced by the launch vehicle, and synchronized the ground track with the reference grid and two verification sites [Bhat 1993]. The initial verification phase involves sensor calibration with *in-situ* data at the verification sites and the assessment of geophysical parameters through flight data analysis, Satellite tracking and laser ranging data are used to perform precision orbit determination and tune the gravity field and other force models used during the remainder of the mission. While this verification process will continue throughout the mission, the initial verification process established the baseline for accurate data processing. This six-month long phase began with the acquisition of the operational orbit and made a smooth transition into the observational phase. Orbit maintenance activities began with initial verification phase and will continue throughout the mission.

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The observational phase Coincides with the satellite design lifetime of three years and the extended observational phase an additional two years. The primary activities during the observational phase are the production and distribution of scientific data and the maintenance of the satellite in a state that allows the data to meet all mission objectives. Scientific data is distributed in the form of Geophysical Data Records (GDR) containing sea surface height, sensor corrections, and geophysical information. To facilitate this mission, the satellite is maintained in a nearly circular, frozen orbit ($e \approx 0.000095$, $\omega \approx 90'$) at an altitude of $\approx$1336 km. and an inclination of $i \approx 66{,}04''$. This orbit provides an exact repeat ground track every 127 revolutions ($\approx$ 9.9 days) and overflies two altimeter verification sites: a NASA site off the coast of Point Conception, ('California (latitude 34,46910 N, longitude 1?().68()81" W), and a CNES site near the islands of 1 .ampione and Lampedusa in the Mediterranean Sea (latitude 35.54649" N, longitude 12.32054 °E).

The TOPEX/POSEIDON operational orbit is given by [Bhat 1993] and was acquired on Sept. 25, 1992. This operational orbit was defined to produce a ground track which repeats every 127 orbits and over flies both verification sites in the absence of non-gravitational perturbations [Vincent 1990A]. An orbit with an exact repeat ground track is obtained by selecting the orbital elements a, i, and Ω so that the sub-satellite ground track repeats with the desired frequency and earth trace density [Karrenberg et. al. 1969]. The original reference orbit was designed [Vincent 1990B] using a 17x17 subset of the GEM-T2 gravity field [Marsh et al. 1989]. Since it was later determined that a 20x20 subset of GEM-T3 [Lerch et. al. 1992] was the minimum field size sufficient for orbit determination with the required accuracy, the reference orbit was tuned prior to launch. The trajectory generation module of the]1'1. double precision trajectory system (DPTRAJ) was used to tune the osculating elements of the reference orbit. DPTRAJ utilizes a predictor-corrector integrator with automatic step size control [Spier, 1971; DPTRAJ, 1971]. This numerical integrator is also used for operational trajectory generation and has the capability of incorporating all relevant perturbing sources including Earth oblateness, luni-solar gravity, atmospheric drag, solar radiation pressure, solid earth tides, polar motion, precession, and nutation.

This paper will describe our success at maintaining the operational ground track. The maintenance of the ground track at the ascending nodes as well as the over flight success will be discussed separately in some detail. Maneuvers which have been performed will be discussed in terms of the targeting strategy used and the successes thereof. Successes and failures of the planned targeting strategies will be examined, and modifications to the maneuver targeting strategies which have been developed and implemented since launch will be described. In particular, changes in the maneuver targeting strategy which were made necessary by an unpredicted along-track satellite-fixed force will be discussed in some detail. The new maneuver targeting strategy which has been developed and is currently in use will be presented.

MANEUVER REQUIREMENTS

Periodic orbit adjustment maneuvers are required to maintain the ground track and ensure that all verification site over flight requirements are met. They must occur on an interference-free basis with scientific data acquisition and precision orbit determination (POD). Specific requirements can be summarized as follows [MSRD 1989]:

1) Maintenance of the operational orbit so that at least 95% of all equatorial crossings at each orbit node are contained within a 2 km longitude band.
2) Maintenance of the operational orbit during the initial verification phases so that it overflies designated locations at two verification sites within ±1 km on at least 95% of the planned over flights.
3) Maintain the eccentricity $e < 0{,}001$. This requirement is automatically met by utilization of the frozen orbit, which is not *per se* a mission requirement.

4) Perform the minimum practical number of orbit maintenance maneuvers during the initial verification phase, with a minimum of 30 days between maneuvers with 95% probability and whenever the 81-day mean 10.7 cm solar flux satisfies $\bar{F}_{10.7} \leq 225$.

5) Orbit maintenance maneuvers are to be performed as nearly as possible to the transition between 127-orbit repeat cycles (±1 rev).

6) The spacing between maneuvers shall be as large as possible during the observational phase of the mission.

7) Maintenance maneuvers are to be performed over land wherever possible. Requirements 5 and 7 are facilitated by defining revolution 1 of the 127-orbit as that revolution which has the most land over flights.

in addition to the above mission requirements, maneuvers may not under any conditions compromise satellite health and safety. This leads to additional restrictions on the timing of maneuvers and the sequence of events required for maneuver implementation. The primary restrictions are due to thermal, power, and satellite attitude control constraints and capabilities

MANEUVER DESIGN

The propulsion module is a mono-propellant hydrazine blow-down system consisting of twelve 1-NT (0.2 lbf) and four 22-NT (5-lbf) thrusters. It was designed to provide sufficient thrust and directional control to meet all orbit adjust and maintenance maneuver requirements, including related attitude control. The 22-NT thrusters and four of the 1-NT thrusters are mounted on the aft facing of the satellite. Each of these thrusters is oriented axially along the x-axis and individually canted to be aligned through the center of mass at the beginning of life when the propellant tanks are full. Large orbit adjust maneuvers (> 400 mm/sec) are performed using the 22-NT thrusters, whereas the much smaller orbit maintenance maneuvers (typically < 10 mm/sec) use a single pair of 1-NT thrusters. The remaining eight 1-NT thrusters are mounted normal to the satellite x-axis to provide attitude control about any of the three body axes.

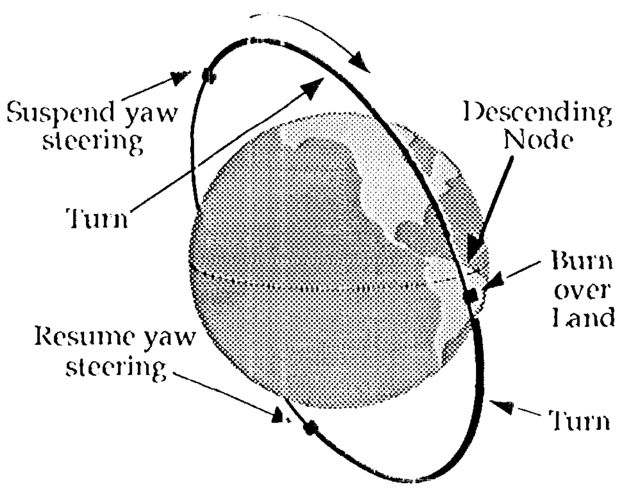
There are two modes of operation in which the on-board software maintains attitude control while in orbit adjust mode. The primary mode is the *open loop firing pattern*, which uses four thrusters. In this mode, an open-loop off-pulsing of axial thrusters to account for anticipated disturbance torques imparted by thruster/center-of-mass offsets is used. All four thrusters (either 22 nt. or 1 NT) must be used in this mode. If the established pitch or yaw angle or rate control limits are exceeded during the burn, orbit adjustment thrusters and the open loop firing pattern are disabled, and the 1-NT thrusters are used to re-establish attitude control. The maneuver is resumed as soon as attitude control is reestablished. In the secondary mode, only two oppositely-located orbit adjustment thrusters are utilized, and no open loop firing pattern is used.

in normal mission mode, the attitude determination and control system maintains the satellite in a stable attitude about all three axes while keeping the z-axis pointed along the local nadir. Near continuous yaw steering about the local nadir and solar panel pitching are utilized to maintain the dominant $28m^2$ solar panel pointed toward the sun for power optimization. The solar array is continuously pitched, with an offset from the true sun line to control the battery charging rate. This control strategy requires the weekly loading of satellite ephemerides into the onboard computer (OBC). This ephemeris load consists of a 42-term Fourier power series expansion of each of the six Cartesian state vector components, fit for a 10-day span, and residuals at 10 minute intervals for the first 30 hours of the load. The OBC recovers this ephemeris by evaluating the Fourier series at four consecutive grid points, adding residuals if available, and performing a four-point Hermite polynomial interpolation. This state vector is then used to define the desired attitude, expressed as a target quaternion. The attitude is maintained by nulling the difference between the actual, as measured by the inertial reference unit, and the target quaternions using reaction wheels.

Fig. 1 illustrates the *turn-burn-turn* sequence used to perform an orbit adjustment maneuver:

1) Suspend nominal attitude control and yaw steering.
2) Slew the satellite to the bias attitude which accounts for the cants of the thrusters and the desired thrust directions.
3) For large maneuvers (requiring the 22-NT thrusters), rotate {he solar array to a 900 or ?70” pitch position as stop solar array pitching.
4) Execute the AV.
5) Return the solar array to sun pointing.
6) Unwind the attitude.
7) Return at titude cent rol to normal mission mode.

Figure 1.
Maintenance maneuver timeline, illustrating the *turn-burn-turn* sequence.



GTARG, which utilizes a mean element propagator including all perturbations that cause significant variations in the satellite ground track, is used for maneuver targeting [Shapiro & Bhat 19931. Two targeting strategies are utilized: (1) maximization of the time between maneuvers (*longitude targeting*); or (2) pre-selection of the minimum time interval between maneuvers (*time targeting*). The mean element propagation includes the effects of Earth renal harmonics through J_{29}, luni-solar gravity, and drag. An orbital average to Jacchia-Roberts model is used to determine atmospheric density. Solar flux ($F_{10.7}$) and geomagnetic parameter (K_p) predictions are based on the daily SESC 3-day and weekly 27-clay outlook. ‘1 'he latest outlooks are combined with observed data to generate a merged 27-da y data set. Missing data are determined by linear interpolation. The solar flux is then extrapolated by repeating the merged data set as required for the prediction span. The 81 -day centered average $\bar{F}_{10.7}$ is calculated from the extrapolated values of $F_{10.7}$. The geomagnetic data are extrapolated at a constant value equal to the average K_p over the first 27 days. A variable mean area model is used to determine the average drag area over an orbit. Maneuvers are modeled impulsively. Error models incorporate the effects of drag prediction uncertainty, orbit determination errors, and maneuver execution errors.

UNPREDICTED ALONG-TRACK SATELLITE-FIXED FORCE

Analysis of tracking data indicates the existence of an unmodeled *along-track force* acting upon the satellite [Frauenholz 1993]. The magnitude of this *along-track force* is equivalent to that of a continuous thrust on the order of micro-newtons. The direction and magnitude are a function of the satellite attitude and β', the angle between the orbit plane and the Earth-sun line. This indicates a satellite-fixed force; suggested causes may be some combination of leakage, outgassing, and a radiative (infrared) force due to a thermal gradient between the two sides of the solar array.

The *along-track satellite fixed force* results in a change of the semi-major axis of approximately of 3-10 cm/day during yaw steering and 25-30 cm/day during the fixed yaw periods after the effects of all other known forces, including drag, are taken into account. During yaw steering, the *along-track satellite fixed force* produces a positive da/dt during periods of negative β' and a negative da/dt during periods of positive β'. The satellite is held at fixed yaw for a 10 to 15 day period around $\beta' = 0$; the yaw orientation is reversed by 180° at the zero point. These $\beta' = 0$ points occur at approximately 56 day intervals. The direction of da/dt has been observed to reverse at the yaw flip. Significant attenuation of the *along-track satellite fixed force* during the fixed yaw period can be made by adjusting the solar array lead or lag angle (subject to satellite power and thermal constraints) to use thermal radiative and solar pressure forces to best advantage. Drag produces a decay ≈ 5 - 15 cm/day, and hence the *along-track satellite fixed force* has the same magnitude of effect upon the orbit as the largest orbital perturbation. The effect of the *along-track satellite fixed force* is described in much greater detail elsewhere [Frauenholz 1993].

MODIFICATIONS '1"0 MANEUVER TARGETING STRATEGY

Earlier analysis [Bhat, Frauenholz and Cannell, 1989] indicated that density estimation errors would strongly dominate the ground track prediction at all times except during the lowest period of solar flux (1",().-, ≈ 'lo). As such, a simple longitude targeting strategy incorporating the ±95% anticipated errors (1 .6453) in all error sources should be used. This strategy biases the targeted ground track eastward so that the 95% envelope is made just tangent to the western edge of the control band.

GTARG was modified to incorporate the alone-track satellite-fixed force via a table look-up model. The table consists of a list of daily da/dt values. The error model was correspondingly mod if id. Starting from the equation (12) of [Frauenholz and Shapiro 1991] $\partial\Delta\lambda/\partial a \approx 3\omega_e t/2a$, where $\Delta\lambda$ *is* the ground track, and introducing a boost of Δa once per orbit for N orbits, then after a time $t \approx NP$,

$$\left.\frac{\partial\Delta\lambda}{\partial a}\right|_{Boost} \approx \frac{3}{2}\frac{\omega_e}{a}\sum_{i=1}^{N} iP \approx \frac{3\omega_e t^2}{8\pi}\sqrt{\frac{\mu}{a^5}} \approx 50t^2(days)\ meters\,/\,meter \quad (1)$$

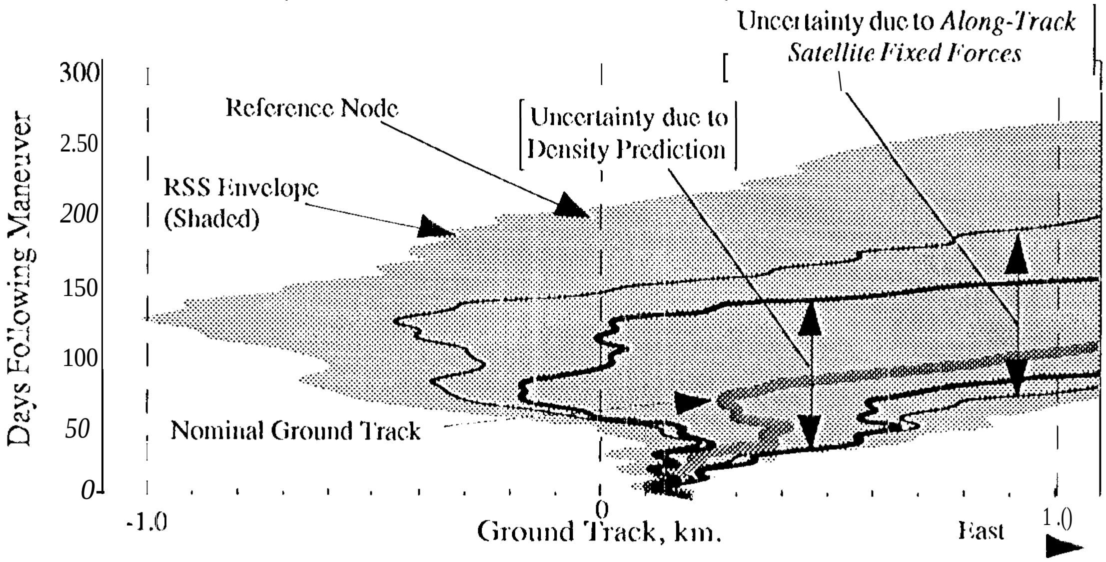
The errors predicted in this way are root-sum-squared with the other error sources to produce the total error model for maneuver targeting.

Naively incorporating the error model of eq. (1) into longitude targeting leads to extremely conservative maneuver design. An example is shown in fig 2, which preforms simple longitude targeting for orbit maintenance maneuver 2 (OMM2), which occurred on Dec 21, 1992. Fig. 2 compares the error contributions due to uncertainties in density prediction along-track forces. The 1<SS envelope includes the contributions of other error sources which are not shown [Bhat, Frauenholz, and Cannell, 1989]. 'The simple longitude targeting strategy produced a AV ≈ 2.6 mm/sec. using σ = 1.7 cm/day for' the uncertainty in the *along-track satellite-fixed force*, and nominal mission error budgets for all other parameters. The large contribution of eq. (1) to the total error budget strongly dominates the targeting after approximate] y 50 da ys. The ground track begins its final t rek east ward at approximate] y day 60 and leaves the control band some ?() days later; yet it is the continued quadratic increase of the error envelope

beyond this time that dominates the targeting. A larger maneuver magnitude, while violating the longitude targeting constraint, would lead to a more even balance between the times at which the edge of the control band are reached.

An additional complication to targeting is introduced by the periods of fixed ya w, I)uring these periods, the magnitude of the *along-track sfllcllife-fixed force* is much larger than during the yaw steering period. Furthermore, the direction of the force is determined by the placement of the ya w-flip, which nominally occurs at β' = 0. For OM M?, the magni t ude of the force could not be predicted in advance, because the orbital geometry was one which had not yet been experienced during the mission.

Figure *2*
Targeted Ground Track and Error Envelope for OM M2.



The paper will explore the alternative targeting strategies which became necessary due to the unexpected *along-track satellite-fixed force*. The new concepts for maneuver targeting will be developed.

SUCCESS OF DENSITY PREDICTION ALGORITHM

The modified Jacchia-Roberts density prediction algorithms as described in [Frauenholz & Shapiro 1991] requires the prediction of solar and geomagnetic data 40 days into the future in order to calculate the present density. Targeting requires predicted densities for periods on the order of 100 to 200 days into the future. Two schemes for generated extended outlooks were compared in the earlier work. Our success at accurately predicting orbital density will be discussed in terms of its relation to maneuver targeting

GROUND TRACK MAINTENANCE

The first part of this section will summarize the maneuvers which have been performed to date in light of the above analysis. These maneuvers are listed in table 1.

Table 1.
Ground track maintenance maneuvers.

OMM #	Date	AV, mm/sec	Aa, meters	Execution Error
1	Oct. 12, 1992	9.43	20.2	3.64 %
2	Dec. 21, 1992	3.15	6.8	-1.46 %

Idealized Ground Track

Ignoring all perturbations with the exception of a constant drag force, and perturbing the exact repeat orbit by a velocity increment ΔV, the ground track has been shown [Bhat, Frauenholz, and Cannell 1989] to follow the parabolic path as shown in fig. 3a. While there are large periodic perturbations on the ground track due to luni-solar gravity, with the exception of periods of extremely low solar activity, the ground track is strongly dominated by drag. The relevant equations will be summarized in the paper.

In fig. 3a the ground track is constrained to remain within a control band defined longitudinally by limits $\Delta\lambda_E$ and $\Delta\lambda_W$. The initial maneuver is performed at the eastern edge of the control band and is targeted so that the parabola is just tangent to, but does not cross, the western edge of the control band. A subsequent maneuver is performed after time T_{max}. The probability that a node falls within an infinitesimal swath of width $d\Delta\lambda$ at $\Delta\lambda$ is proportional to the time spent within the infinitesimal swath at $\Delta\lambda$. The probability density function will be shown in the paper to be given by

$$P(\Delta\lambda)d\Delta\lambda = \frac{1}{2\sqrt{(\Delta\lambda_E - \Delta\lambda_W)(\Delta\lambda - \Delta\lambda_W)}}d\Delta\lambda \qquad (?)$$

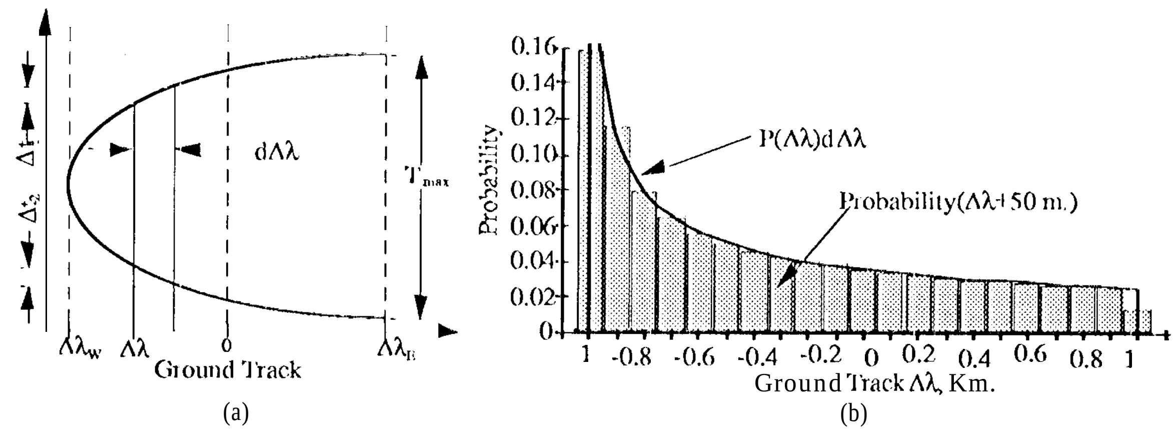
The mean value $\bar{\lambda}$ and standard deviation σ are

$$\overline{\Delta\lambda} = \frac{1}{3}(2\Delta\lambda_W + \Delta\lambda_E) \qquad (3)$$

$$\sigma = \frac{2}{3\sqrt{5}}(\Delta\lambda_E - \Delta\lambda_W) \approx 0.298(\Delta\lambda_E - \Delta\lambda_W) \qquad (4)$$

Thus the node distribution will be heavily skewed towards the western side of the control band, with a mean value one third of the way into the band. For the idealized TOPEX/POSEIDON ground track, $\lambda_E = 1$ km. and $\lambda_W = -1$ km. The corresponding density function is shown in fig. 3b. The true nodal distribution after multiple maneuvers will be more complex than the simple form given in equation *(2)*. This is because maneuvers will not always occur precisely at $\Delta\lambda_E$, the ground track will not always turn around precisely at $\Delta\lambda_W$, and the ground track will deviate from the simple parabolic shape of fig 3 due to luni-solar and satellite-fixed forces.

Figure 3.
Statistical distribution of ground track: (a) idealized ground track geometry; (b) probability density function.



Observed Ground Track

The TOPEX/POSEIDON satellite ground track since entering the observational orbit will be discussed in this section. Fig. 4 shows the ground track as measured at the nodal crossings, and fig. 5 shows the statistical distribution of the nodes. Deviations from the ideal ground track of fig. 3 are primarily due to the variability of the drag force due to solar and geomagnetic activity, luni-solar gravity, central body gravitational harmonics, and body-fixed forces. Other higher order perturbations, such as solar radiation pressure, thermal gradients and satellite outgasing may also have an effect. The paper will explain the reasoning for performing maneuvers at the times shown, rather than at the ideal times T_{max}, and will discuss why the ground track deviates from the ideal theory of fig. 3.

Figure 4
TOPEX/POSEIDON ground track. The vertical grid lines correspond to cycle boundaries.

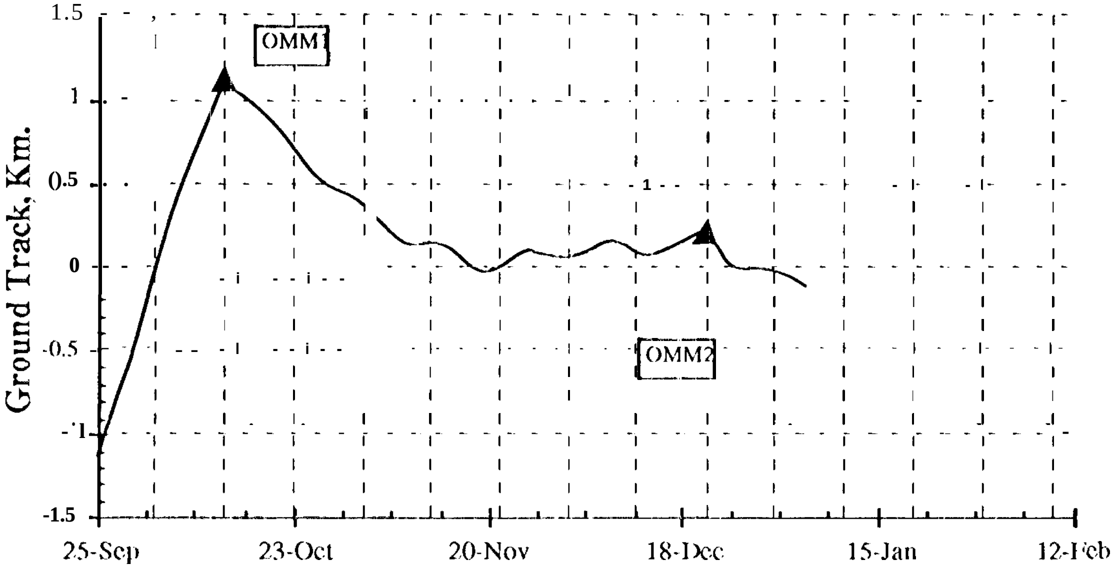
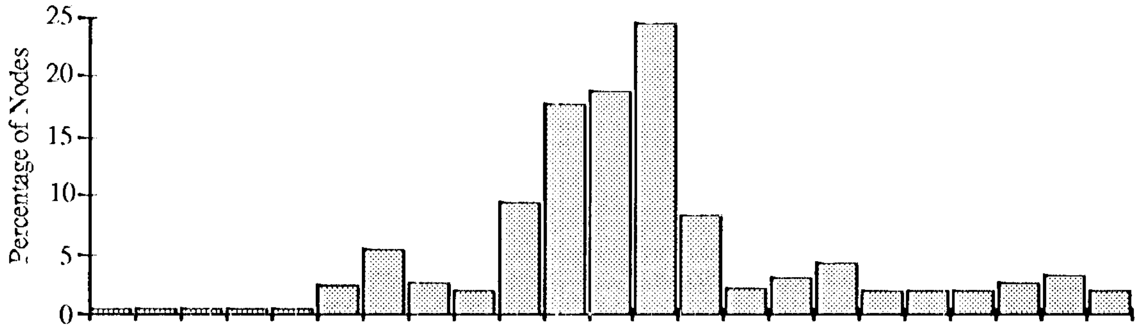


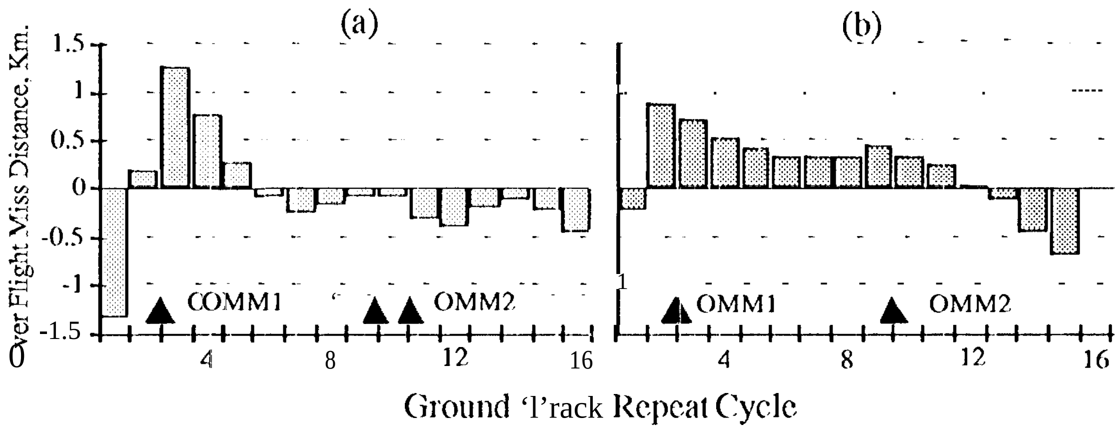
Figure 5
Statistical distribution of node crossing longitudes.



VERIFICATION SITE OVERFLIGHTS

The closest approach x of the ground track to the verification site will be shown in the paper to satisfy $x \approx 0.72\Delta\lambda \approx 0.87d$, where d is the longitudinal miss distance, measured at the site, and $\Delta\lambda$ is the longitudinal miss measured at the equator. Thus maintaining the ground track at the nodes ensures that the requirement at the verification sites is also met. In fact, the ground track will be maintained nearly 30% more tightly at the sites than at the nodes. Furthermore, the longitudinal offset at the site exceeds the distance of closest approach by approximately 13%. Additional complications are introduced if there is a inclination error, which will be discussed in the paper. The longitudinal offsets of the ground track at the verification sites is shown in fig. 6. This will be discussed in light of the earlier analysis.

Figure 6
TOPEX/POSEIDON Ground Track at the Verification Sites: (a) NASA; (b) CNES. The longitudinal offset as measured at the verification sight is shown.



SUMMARY

Since attaining the operational orbit, the ground track has been maintained with $\mu \approx 0.15$ km and $\sigma \approx 0.30$ km. Over 97% of the more than 1600 nodal crossings which occurred during this time have been within the ±1 km reference bandwidth. The success of meeting mission requirements using the methods we have developed and implemented will be summarized. A simple longitude targeting strategy incorporating the ±95% anticipated errors as a window on the predicted post-maneuver ground track, was designed prior to launch. This strategy was evaluated in orbit, and had to be modified to incorporate the effects of an unexpected satellite-fixed along-track force. While new strategies have been developed and successfully used in-orbit, the primary open question that remains is how well these strategies will work in the extremely low solar-activity regime of the approaching solar minimum. Earlier analysis indicated that a heavier reliance on a time-targeting strategy would be necessary. Whether the earlier conclusions are still valid, or need to be reexamined in light of the effect of the satellite-fixed along-track force is a subject for further analysis.

REFERENCES

[Bhat, 1991] Bhat R. S., "TOPEX/POSEIDON orbit Acquisition Maneuver Design," AAS 91-514, 1991 *AAS/AIAA Astrodynamics Specialist Conference,* Durango, Colorado, August 19-22,1991.

[Bhat, 1993] Bhat, R.S., "TOPEX/POSEIDON Operational Orbit Acquisition", AAS/AIAA Astrodynamics Conference, Victoria, BC, Canada, Aug. 16-19,1993, (see submitted abstract).

[Bhat, Frauenholz, and Cannell, 1989] Bhat, R.S. , Frauenholz, R. B., and Cannell, P. E., "TOPEX/POSEIDON Orbit Maintenance Maneuver Design," AAS 89-408, *AAS/AIAA Astrodynamics Specialists Conference*, Stowe, VT, August 7-10, 1989.

[Carlisle, 1991] Carlisle, George, A. DiCicco, H. Harris, A. Salama, M. Vincent, *TOPEX/Poseidon Project Mission Plan*, Jet Propulsion Laboratory, JPL D-6862, rev. C, Aug. 1991 (Internal Document).

[DPTRAJ, 1971] *DPTRAJ/ODP User's Reference Manual*, Jet Propulsion Laboratory, JPL D-263, Oct. 15,1971 (Internal Document).

[Frauenholz and Shapiro 1991] Frauenholz, R. B., and Shapiro, B.E., "The Role of Predicted Solar Activity in TOPEX/POSEIDON orbit Maintenance Maneuver Design," AAS 91-515, *AAS/AIAA Astrodynamics Specialists Conference*, Durango, CO, August *19-2?, 199].*

[Frauenholz, 19'93] Frauenholz, R. B., Hamilton, T. W., Shapiro, B.E., and Bhat, R. S., "The Role of Anomalous Satellite Fixed Forces on TOPEX/POSEIDON Orbit Maintenance," AAS/AIAA Astrodynamics Conference, Victoria, BC, Canada, Aug. 16-19, 1993 (see submitted abstract).

[Karrenberg, 1969] Karrenberg, H. K., Ltwin, E., Luders, R. D., "Orbit Synthesis", *J. Astr. Sci*, XVII, 3, pp129-177, Nov-Dec. 1969.

[Lerch, 1992] Lerch, F. J., et. al., *Geopotential Models of the Earth from Satellite Tracking, Altimeter and Surface Gravity Measurements: GEM-T3, GEM-T3S*, NASA Technical Memorandum 104555, Jan. 1992.

[Marsh, 1989] Marsh, J. G., et. al., *The GEM-T2 Gravitational Model*, NASA Technical Memorandum 100746, Oct. 1989.

[MSRD, 1989] *TOPEX/POSEIDON Project* Mission *and Systems Requirements*, Jet Propulsion Laboratory JPL D-5901, April 1989 (Internal Document).

[Shapiro and Bhat, 1993] Shapiro, Bruce, and Bhat, Ramachandra, "GTARG - The TOPEX/POSEIDON Ground Track Maintenance Maneuver Targeting Program", Paper AIAA 93-1129, *AIAA Aerospace Design Conference*, Irvine, CA, Feb. 16-191993,

[Spier, 1971] Spier, Gerd W., *Design and Implementation of Models for the Double Precision Targeting Program (DPTRAJ)*, Jet Prepulsion Laboratory, Technical Memorandum 33-451, April 15, 1971 (Interns] Document).

[Vincent, 1990A] Vincent, M. A., "The Inclusion of Higher Degree and Order Gravity Terms in the Design of a Repeat Ground Track," AI AA-90-2899 -CP, *AIAA/AAS Astrodynamics Conference*, Portland, Oregon, August 1990.

[Vincent 1990B] Vincent, M. A., *TOPEX/POSEIDON Orbit Characteristics*, Jet Propulsion Laboratory, JPL D-7511, June 1990 (Internal Document).